IAC–2017–C1.7.11

Mission Design and Optimal Asteroid Deflection for Planetary Defense

Bruno V. Sarli
Associate Researcher, Catholic University of America, bruno.victorinosarli@nasa.gov

Jeremy M. Knittel and Jacob A. Englander and Brent W. Barbee
Aerospace Engineer, NASA Goddard Space Flight Center, jeremy.m.knittel@nasa.gov and jacob.a.englander@nasa.gov and brent.w.barbee@nasa.gov

Planetary defense is a topic of increasing interest. Many reasons were outlined in "Vision and Voyages for Planetary Science in the Decade 2013-2022". However, perhaps one of the most significant rationales for asteroid study is the number of close approaches that have been documented recently. A planetary defense mission aims to deflect a threatening body as far as possible from an Earth-impacting trajectory. The design of a mission that optimally deflects an asteroid has different challenges: speed, precision, and system trade-offs. This work addresses such issues and develops an efficient transcription of the problem that can be implemented into an optimization tool. This allows for a broader trade study of different mission concepts with medium rather than low fidelity. Such work is suitable for preliminary design. The methodology is demonstrated using the fictitious asteroid 2017 PDC. The complete tool is able to account for the orbit sensitivity to small perturbations and quickly optimize a deflection trajectory. The speed at which the tool operates allows for comparisons between different spacecraft hardware configurations. Finally, key deflection dates and mission strategies are identified for 2017 PDC.

I. Introduction

As missions to asteroids become more common and the risk of an asteroid impact becomes better understood, as demonstrated by the Chelyabinsk meteor,[1] planetary defense becomes a topic of increasing importance. A deflection mission to an asteroid has the clear objective of safely driving the body as far as possible from a trajectory which impacts the planet. This paper will focuses on two of the most studied deflection mechanisms for short warning times, the kinetic impactor (KI) and the nuclear explosion device[2] (NED). Past studies have developed different models for quantifying deflection. Main contributions to this area rely on analytical approximations of the closest approach state[3] or n-body propagation of the asteroid's post-deflection orbit.[4,5] Optimization strategies thus far utilize different concepts: direct methods,[6,7] indirect methods,[8] orbit mapping[9] and orbital approximations.[10] The method developed here incorporates the trajectory design of the spacecraft with a simple set of two-body propagations to define the asteroid's post-deflection path. This provides a fast and simple approximation with medium accuracy, suitable for preliminary mission design.

The objective of this paper is to outline a new modeling technique for optimal asteroid deflection that can be incorporated into the spacecraft trajectory design process. This model uses real asteroid ephemerides and calculates the post-deflection trajectory by applying a Lambert fit to correct for the asteroid's velocity, at the time of deflection, using Keplerian dynamics. Once the natural asteroid path is adjusted for two-body dynamics, the post impact trajectory is constructed using three point parallel shooting. Two of the propagations take place in the heliocentric frame. The first is in the forward time direction, originating at the initial deflection. The second is in the backward time direction from the asteroid's entrance into Earth's sphere of influence (SOI). The third propagation takes place with Earth as the central body, forward in time from the entrance into the SOI. This method allows for rapid calculation in medium fidelity achieved through the use of real ephemerides and a Lambert corrector. The framework is therefore rapid enough to be suitable for a trajectory optimization tool, but does not need excessive approximations which would render the result to be of little worth. Inside an optimization, the close approach distance can be used as the objective function, and the spacecraft trajectory can be designed for an optimal deflection.

A mission concept is presented utilizing the asteroid impact scenario presented by the fictitious small body, 2017 PDC.[11] Solutions target an optimal deflection of the body utilizing the two aforementioned

methods (KI and NED). Low-thrust trajectories were selected for the design due to the complexity and challenge they pose to an optimizer. The optimization tool also allows for the inclusion of real mission constraints[13] and a trade-off between mission parameters,[12] such as launch vehicle, propulsion system and the launch epoch. Lastly, the post-deflection asteroid state was propagated in high fidelity to confirm that an optimal deflection was obtained with the medium fidelity model.

This paper will overview vital design considerations for a deflection mission (Section 2), the new model for post-deflection dynamics (Section 3), a summary of the assumptions for the sample study (Section 4), the resulting trajectory designs (Section 5) and finally the study's conclusions (Section 6).

II. Design Considerations

II.i Deflection

Asteroids can be deflected by forcefully altering their velocity vector. This can be accomplished with many different methods, but the most studied for a short ($<\sim 7$ year) warning time are a kinetic impactor (KI) and a nuclear explosive device (NED).

The KI concept imparts a change in velocity to an intercepted asteroid by transferring energy using a colliding spacecraft's momentum. On the other hand, a NED changes the target's velocity using the force of the explosion. Because of the massive energy release generated by a nuclear explosion, this concept is more mass efficient than a KI.

Deflection concepts that transfer a considerable amount of energy in a very short time, such as KI and NED, can generate debris that escape the body and become secondary threats. In other cases, the energy transfer can break the asteroid into multiple large pieces. Depending on the size, mass and material the asteroid is made of, its cohesive force might not be sufficient to sustain a medium nor high energy impact. Such an event could break the asteroid apart and generate two or more objects with similar Earth-colliding orbits.

Therefore, the change in velocity imparted to the asteroid needs to have an upper limit. This ceiling is typically scaled as a certain percentage of the velocity required to escape the asteroid's gravity field.

This translates to a maximum allowed impact velocity for KI missions. Similarly, a NED's force can be sized to generate the desired velocity change, or the stand off distance for detonation can be varied to achieve the same effect.

The relative speed between the spacecraft and the asteroid is an important mission driver beyond the need to achieve a targeted asteroid velocity change. The navigation system on a spacecraft can only handle a range of approach speeds and still guarantee the correct impact conditions. Regardless of the deflection method, the velocity that the spacecraft can impart on the asteroid is several orders of magnitude smaller than the body's heliocentric velocity; therefore, the deflection must occur significantly before Earth impact.

Deflection missions must be launched years, if not decades, in advance for the change in the asteroid's orbit to sufficiently deviate from the original orbit and safely transit past Earth without an impact.

Each deflection method has a range of applicability, determined by the time available before impact and the mass of the target. Figure 1 outlines the general range for different deflection strategies with respect to mission time and target size. The KI strategy is appropriate for the range of most Potentially Hazardous Asteroids (PHA).[14] However, this method is not efficient for larger asteroids or when time for deflection is short. In such cases, a NED is the only feasible option.

This type of device is effective because it can provide a significant quasi-instantaneous velocity variation, Δv, to the target. Nevertheless, transferring large amounts of energy to an asteroid is not always going to generate a deflection.

Depending on size, mass, and material, an asteroid's cohesive forces may not be sufficient to sustain a medium or high impact. Such an event would break the asteroid apart and generate two or more objects with similar orbits on a collision course with Earth. A second point of concern in high energetic deflections is the ejecta generated. It is important to limit it to an estimation of the escape velocity to prevent potential secondary impacts from these debris. There is a specific class of problem, however, that the time available and the target characteristics will yield no satisfactory deflection. In such cases disruption of the target may be the only alternative. Obliteration of the target is a complex calculation and it is out of the scope of this study.

II.ii Optimization

As mentioned above, the velocity change that the mission imparts on the asteroid is extremely small compared to the target's original velocity. An optimizer, therefore, needs to be able to propagate with sufficient precision to not numerically lose the effect

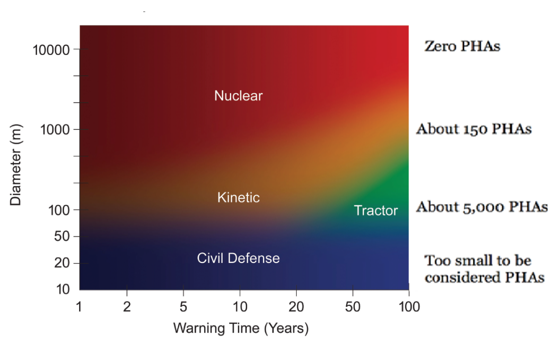


Fig. 1: The four types of mitigation and their regimes of primary applicability[14]

of the deflection. Propagation with precision of the same order of magnitude as the velocity change will result in an identical orbit before and after deflection.

A similar precision error can occur during the change of reference frame from Heliocentric to geocentric. The velocity of Earth in the heliocentric frame is much larger than the differences between candidate Earth approaching post-deflection asteroid trajectories. If the precision of the frame transfer is not sufficiently high, then the deflection could be shown to be effectual when in fact it would not have been successful in a higher-precision numerical simulation.

Given the requirements on sufficient numerical precision, it bears mentioning why the propagations are not performed entirely using full n-body models. This method would certainly alleviate the numerical precision concerns, but it is better suited for later stages of the mission design due to the long computation time required. N-body propagation is currently infeasible for early mission concept studies where many dozens of mission configurations are tested and trade studies are conducted, each requiring a full trajectory optimization and the associated thousands or millions of propagations.

On the other end of the numerical precision spectrum, analytical approximations could be used to evaluate the difference between the original orbit and the deflected orbit in a phase-free problem. There are so many approximations in such a calculation, however, that the result is only useful as an upper and lower bound for what the mission can achieve. A real deflection mission requires consideration of timing and deflection angle and these are both ignored in phase-free calculations. It is unlikely that a real mission would be able to launch at the exact moment required to reach the phase-free optimum, as it requires an idealized geometry between the Earth and the asteroid, which infrequently exists.

III. Deflection Model

The most crucial aspect of a rapid and accurate deflection optimization routine is the modeling of the asteroid's post-defection trajectory. The time-scale of both KI and NED deflections is sufficiently short that the change in the target's velocity can be modeled as an instantaneous Δv applied at a certain epoch.

The KI's momentum transfer is dependent on the spacecraft's arrival velocity and mass, as described in Eq. 1.

$$\Delta \mathbf{v} = \mathbf{v}_\infty \beta \frac{m_{S/C}}{m_{S/C} + m_{asteroid}} \quad [1]$$

where, $\Delta \mathbf{v}$ is the velocity change vector imparted on the asteroid by the spacecraft's impact, $\mathbf{v}_\infty$ is the spacecraft's arrival velocity vector relative to the asteroid, β is the momentum enhancement factor which encompasses the plasticity of the impact (β can be 1 or higher), and $m_{S/C}$ and $m_{asteroid}$ are the spacecraft and asteroid masses, respectively. To make a conservative approximation of the impact, $\beta = 1$ is used here. A higher β would be beneficial because it would mean that the impact generates a higher Δv, which, in turn, provides a higher deflection.

The momentum enhancement factor is not yet well understood for lack of empirical data from spacecraft test missions. Various computer simulations have been attempted by researchers in an effort to predict likely ranges of β values for representative asteroids. These simulations generally treat the spacecraft as a simple solid sphere, while varying the properties of the asteroid. As such, these simulations principally investigate possible effects of the asteroid properties on β. Current results suggest that β is likely in the range of 1 to 5,[15] and possibly higher for dense monolithic asteroids. However, dense monolithic asteroids are thought to be very uncommon in nature, and none have been observed to date. Most asteroids are thought to be porous, possibly to the point of being "rubble piles" only held together by self-gravity and electrostatic forces. Simulations exploring the effects of spacecraft design on β are much more computationally complex, and limited results are available in the literature. The computational complexity arises from the need to simulate the collision of a detailed spacecraft computer aided design (CAD) model with an asteroid. Weaver, et al (2015)[16] present results for spacecraft design effects on forming craters on asteroids. The results indicate that a more pointed, dense design for the spacecraft (like a penetrator) is more effective at crater formation than a typical "box" shape

spacecraft that has significant empty space and a lack of concentrated mass. While Weaver, et al (2015) did not explore β, their results seem to suggest that a penetrator-type design for a spacecraft might improve β, by virtue of deeper and more efficient crater production.

The use of an explosive device adds a new level of complexity to the problem. To avoid asteroid ejecta or asteroid break-up, the incident radiation on the target's surface needs to be constrained, which results in a balance between the size of the explosive and the standoff detonation distance. For cases where the deflection is not possible an obliteration of the asteroid is the best option. A surface or subsurface detonation can also be considered as the last resource. A detailed calculation for the size of the nuclear explosive and the distance at which it should be detonated is not included in this work as it is protected, proprietary information. Instead, the effect of the NED will simply be a Δv applied to the asteroid's velocity. The magnitude is fixed for a given optimization run and is pre-calculated using the aforementioned proprietary model. A representative value of Δv is presented in Section 4.

Qualitatively, the objective of the optimization is to maximize the close approach distance as the asteroid passes Earth. This can be modeled as the asteroid's radius of periapse in the geocentric two body frame,

$$J = r_p = a(1-e) \qquad [2]$$

where, a and e are the semi-major axis and eccentricity of the Earth-spacecraft two body problem, respectively. The optimization tool then is free to select the optimal time and direction of impact to maximize Eq. 2.

This work makes use of NASA's Evolutionary Mission Trajectory Generator (EMTG) software package; more information and details of this optimization tool and how it model's the pre-impact portion of the spacecraft's trajectory can be found in Englander, et al (2015).[17] The model of the asteroid's trajectory post-impact is the main focus of this work. Upon impact or detonation, the spacecraft is assumed to be vaporized. For cases where the NED is released during the flyby, the spacecraft is assumed to no longer be in the vicinity of the target. The post-deflection trajectory is divided in two phases: the heliocentric transfer, where second order effects play a central role in defining the asteroid's approach to Earth, and the close approach, where Earth's gravity is the dominant acceleration. Earth's SOI is used as the natural divider between the two phases. A simple and fast Keplerian propagator[18] is employed to calculate the asteroid's post-deflection orbit. This, however, does not account for perturbations and, therefore, does not perfectly track a real orbit. In fact, a Keplerian propagation of the nominal asteroid's trajectory would incorrectly place the threatening object far from the planet at close approach time. Moreover, the deflection Δv is several orders of magnitude smaller than the target's velocity. Both of these issues suggest the need for more precise propagation, however this would be prohibitively time consuming.

To avoid this, two correctors are employed. First, a Lambert fit is used on the asteroid's nominal orbit (no deflection) to obtain a two-body version of the body's velocity that would result in the same Earth SOI crossing position as the real orbit. The Lambert fit's starting point is at the time of the spacecraft impact or detonation and the final point is the time of the Earth's SOI crossing. This results in an artificial velocity vector for the asteroid at the time of deflection that guarantees the same SOI crossing point. Because the epoch of the deflection impact is selected by the optimization algorithm, this Lambert fit must be done with every objective function evaluation. Second, in order to guarantee that the two-body non-deflected asteroid orbit has the same r_p as in an n-body propagation, the Earth's velocity is also corrected. This guarantees that the asteroid's hyperbolic arrival velocity is the same and generates the same close approach position. In both cases the SOI crossing time can be found using a bisection method.

Once the two-body regime is adapted to provide meaningful solutions, the post-deflected trajectory can be modeled with three point shooting, as shown in Fig. 2. The boundary conditions for the propagations are: the asteroid's state just after the deflection (forward in time), the asteroid's Earth SOI crossing state in heliocentric coordinates (backward in time) and the asteroid's Earth SOI crossing state in geocentric coordinates (forward in time). The control variables are:

$$\mathbf{U} = [\Delta\mathbf{v}, \mathbf{v}_\infty, \alpha, \delta]^T \qquad [3]$$

where, $\Delta\mathbf{v}$ is provided by the deflection strategy used (KI or NED), $\mathbf{v}_\infty$ is the asteroid hyperbolic excess velocity at the Earth's SOI crossing and α and δ are respectively the azimuth and elevation of the SOI crossing point with respect to Earth's center.

The constraints associated with Eq. 3 are:

$$F = \begin{bmatrix} \mathbf{x}^+_{Match} - \mathbf{x}^-_{Match} = 0 \\ -1 < \frac{\mathbf{v}_\infty \cdot \mathbf{r}_{SOI}}{v_\infty r_{SOI}} < 0 \end{bmatrix} \qquad [4]$$

where, $\mathbf{x}^+_{Match}$ and $\mathbf{x}^-_{Match}$ are the asteroid state just before and after the heliocentric Keplerian propagations. Again, these propagations take place over one half of the time from deflection until SOI crossing. The net duration of the heliocentric phases is calculated by the bisection method. The frame transfer is then calculated as: $\mathbf{r}_{SOI} = r_{SOI} \left[\cos\alpha\cos\delta, \quad \cos\alpha\sin\delta, \quad \sin\delta \right]^T$, $r_{SOI} = a\left(\frac{\mu_{Earth}}{\mu_{Sun}}\right)^{0.4} = 0.924\times 10^6$ km is the size of the Earth's SOI, μ_{Earth} and μ_{Sun} are respectively the standard gravitational parameter of the Earth and Sun.

The first element of F guarantees the continuity of the heliocentric phase and the second element guarantees that the arrival SOI direction is entering and not exiting the system.

Note that this transcription gives direct control over the objective function (Eq. 2) by converting the states from Cartesian to Keplerian elements (Eqs. 5 to 8), which is a desired characteristic for such problems resulting in improved convergence behavior.

$$\varepsilon = \frac{v_\infty^2}{2} - \frac{\mu_{Earth}}{r_{SOI}} \quad [5]$$

$$a = -\frac{\mu_{Earth}}{2\varepsilon} \quad [6]$$

$$\mathbf{e} = \frac{\left(v_\infty^2 - \frac{\mu_{Earth}}{r_{SOI}}\right)\mathbf{r}_{SOI} - \left(\mathbf{r}_{SOI}\times\mathbf{v}_\infty\right)\mathbf{v}_\infty}{\mu_{Earth}} \quad [7]$$

$$e = \|\mathbf{e}\| \quad [8]$$

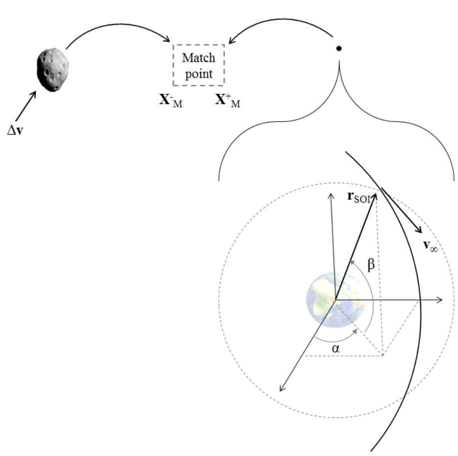


Fig. 2: Asteroid post-deflection three point shooting

In this work, we focus on spacecraft that use solar electric propulsion. This type of low-thrust propulsion system is capable of very high specific impulse. However, trajectory design is challenging as the thrust arcs are long and there are many control variables to select. Furthermore, the engine's performance varies with available power, and therefore the distance from the Sun, as well as the logic used to switch amongst multiple thrusters.

Asteroids are sufficiently small that their gravity can be ignored. Two basic arrival types are considered: high-relative velocity and rendezvous. Rendezvous, in general, requires greater propellant by necessitating that the spacecraft matches the asteroid's velocity at arrival. This is especially difficult for targets with high eccentricity and high inclination such as comets and certain asteroids. On the other hand, there is a maximum approach velocity, due to limitations of on board navigation systems and assumed to be 10 km/s here. Furthermore, a high-velocity deflection fixes many parameters of the asteroid approach, and creates directions which a KI or NED cannot impart momentum as such from behind the asteroid in the approach direction. Even if the spacecraft only performs a flyby of the asteroid, and detonates an NED from a close approach, this limitation is not removed.

F^* in Eq. 10 represents an additional constraint if a flyby is used with an NED. This condition guarantees that the detonation cannot happen behind the target.

$$\hat{\sigma} = \arcsin\left(\frac{r_{ast}}{d_{detonation} - r_{ast}}\right) \quad [9]$$

$$F^* = [\sigma - \hat{\sigma} > 0] \quad [10]$$

where, σ is the angle between the spacecraft arrival velocity and the NED's Δv, $\hat{\sigma}$ is the angle that defines the portion of the asteroid where the detonation cannot occur, r_{ast} is the asteroid radius, and $d_{detonation}$ is the pre-defined standoff detonation distance.

The rendezvous option provides a comfortable condition for the navigation system (low approach speed), but feasible trajectories are far fewer with a reasonable available propellant budget. This option can only be considered for NED deflections, as the kinetic impact would have no effect at a relative velocity at or near 0 km/s.

IV. Example Application

2017 PDC is a fictional asteroid which presents an impact scenario developed by NASA Center for Near Earth Objects (CNEOS) for the purposes of studying planetary defense missions and protocols for the 2017 IAA Planetary Defense Conference.[20] In the hypothetical impact scenario, a potentially hazardous

asteroid is discovered on March 6, 2017. The first probability estimate of an Earth impact is about 1 out of 40,000. The asteroid is assumed to have an approximate diameter of 385 meters with a density of $2.6g/cm^3$ (mass is 7.768804e10 kg). After an observation campaign, the impact probability rose to 1%. Later, there is confirmation of an Earth impact on July 21, 2027.

Little information is known about the threatening body and very few constraints were imposed by the problem's creators. This allows for a trade study between different mission options. It is assumed that space assets currently available will also be at the mission's disposal. Different system options can be considered for the mission design, such as: launcher, launch date, deflection method, trajectory type, and propulsion system. The trajectories developed in this work take into account realistic mission constraints that are outlined in Table 1.

In addition to the trajectory constraints, the mission is required to perform a survey of the target asteroid before the deflection. A survey mission prior to a deflection is important to characterize 2017 PDC's topology, size, spin state and orbit as precisely as possible to increase the likelihood of successful targeting, relative navigation, and deflection.

Two launchers are considered in the study: Atlas V and Delta IV Heavy. More powerful future launch systems that are under development and intended to be operational in the intervening years would only improve the solution space. The selection of a launch vehicle has implications on the trade off between launch mass and excess escape velocity. In general, a high departure velocity is beneficial to kinetic impactors as it can provide a larger deflection by allowing the vehicle to reach the asteroid earlier and with more velocity. It is important to note that higher arrival velocities do not necessarily ensure a better deflection as impact specific energies are limited by the target's composition and density.

A value of 100 J/kg is assumed here as an upper bound for the asteroid's cohesion energy. The spacecraft launch mass is left free to vary as one of the optimization parameters as a function of the selected launcher's capability.[21] Nevertheless, the arrival mass is also bounded to guarantee a viable size for the spacecraft; a minimum bound is placed on the spacecraft arrival mass of 1900 kg for NED and 500 kg for KI and the surveying spacecraft. These mass limitations account for the deflection devices, subsystems and propellant margin.

The spacecraft mass will vary during the trajectory when propellant is used. Two NEXT TT11 high-thrust engines are used as the propulsion system, with a throttle logic algorithm which prefers the minimum number of thrusters to be on while not wasting any available power. A duty cycle of 90% is used to leave margin for missed thrust and trajectory corrections due to secondary effects. At a distance of 1 AU from the Sun, the power system generates 20 kW, with a $1/r^2$ relationship as distance varies. The spacecraft bus requires .8 kW of power at all times.

V. Trajectory Design

V.i Peak Deflection Points Survey

Prior to performing a full trajectory optimization, a survey was made to identify peak deflection points using high-fidelity propagation. These will be used as a comparison for the results of the trajectory optimization to ensure that the epoch of deflection is indeed optimal, as well as the deflection angle. The high-fidelity model incorporates the gravity effects of all the solar system's planets, moons and Pluto. The same model and ephemeris is used to generate the unperturbed 2017 PDC orbit.

To generate these high-fidelity results, the unperturbed asteroid is given a Δv in every direction (discretized at one degree increments) and the best solution is recorded. The magnitude of the impulse is fixed to 1.8 cm/s, which is the assumed capability of the NED. This value can be considered a maximum, as a KI does not have sufficient mass to generate as much Δv as a NED.

This process is repeated for the 3000 days prior to the asteroid's impact with Earth. Figure 3 shows the results of the survey. Note that there are two deflection peaks. The first is on February 26, 2020 (2502 days before close approach) and the second is on March 12, 2024 (1448 days before close approach). With a NED capable of the velocity increment described above, the peaks generate a deflection of 3.62 and 1.32 Earth radii, respectively.

V.ii Optimal Solutions

In a single spacecraft mission concept, the same vehicle has to perform both survey and deflection. Figures 4 and 5 show solutions for flyby and rendezvous. Note, the flyby option can only deflect the asteroid after the second peak, where the deflection is too small to avoid an Earth impact in both options, NED and KI. Utilizing the rendezvous option, the spacecraft arrives after the first peak, but before the second peak. This allows ample time for the survey campaign and a precise detonation as the timeframe

Table 1: Mission constraints

Constraint	Value	Reason
Launch date	after Aug. 1, 2019	2 years after the asteroid's probability of Earth impact rises to 10%.
Launch declination	± 28.5	Declination bounds for the Kennedy launch complex.
Asteroid encounter phase angle	≤ 120	Upper limit to have enough of the asteroid illuminated for the spacecraft's terminal guidance system.
Sun minimum distance	0.7 A.U.	Lower limit for the spacecraft design to handle the more aggressive thermal and radiation environments.
Sun maximum distance	3.5 A.U.	Upper limit to design a large spacecraft (complicated) enough to handle power generation and Earth communications at greater distances.
Earth Angle at asteroid encounter	≥ 3	Lower limit for the Deep Space Network to guarantee a viable RF link with the spacecraft.

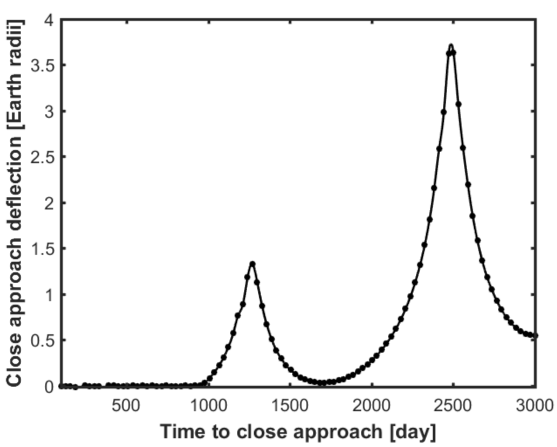


Fig. 3: 2017 PDC peak deflection dates

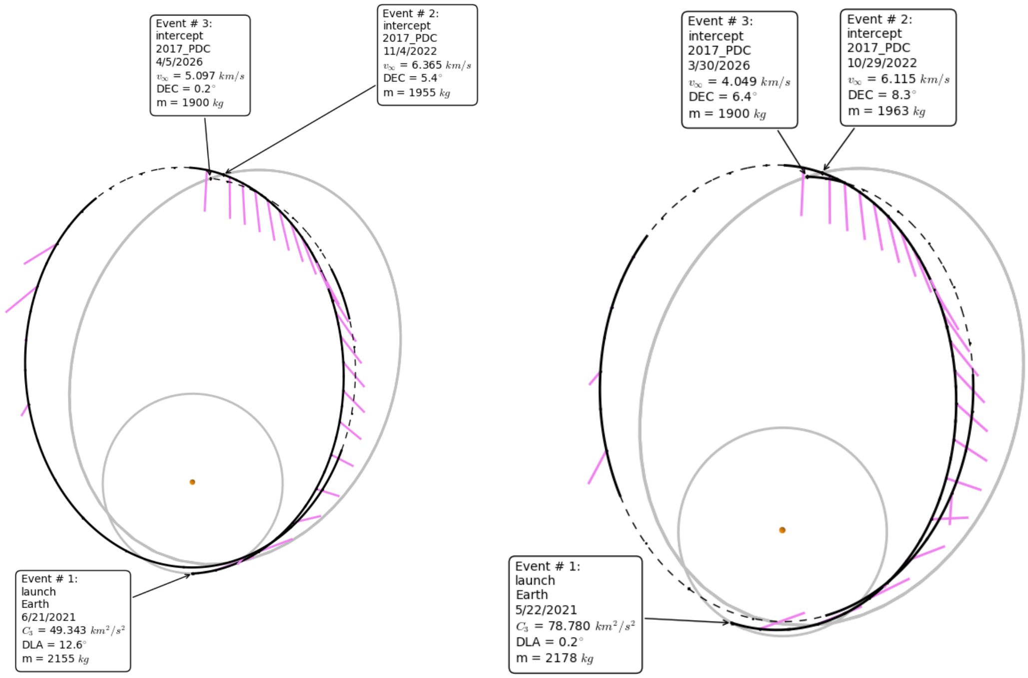
of the second peak nears. Clearly, this particular option is only suited for a NED and it would not be effective with a KI.

With a double spacecraft concept, the first vehicle needs to survey the asteroid before the second spacecraft is launched. This option requires only high-velocity approach scenarios to be explored as rendezvous solutions can be successful with a single vehicle. Figures 6 and 7 show, respectively, solutions for the survey and high-velocity deflection. Both solutions show a valid mission profile with the deflection happening at the second peak. This is therefore suitable for both KI and NED strategies.

The solutions found by the optimizer with the double spacecraft concept matches the peaks found in the survey, which shows that the deflection model correctly represents the close approach in high-fidelity. The magnitude of the deflection calculated by the optimizer in the second peak also matches closely the values obtained with the high-fidelity model: 1.48 Earth radii in March 12, 2024. Making a total difference of $1.48 - 1.32 = 0.16$ Earth radii; a 12% relative error. No feasible solution was obtained to match the first peak, due to launch date constraints.

VI. Conclusions

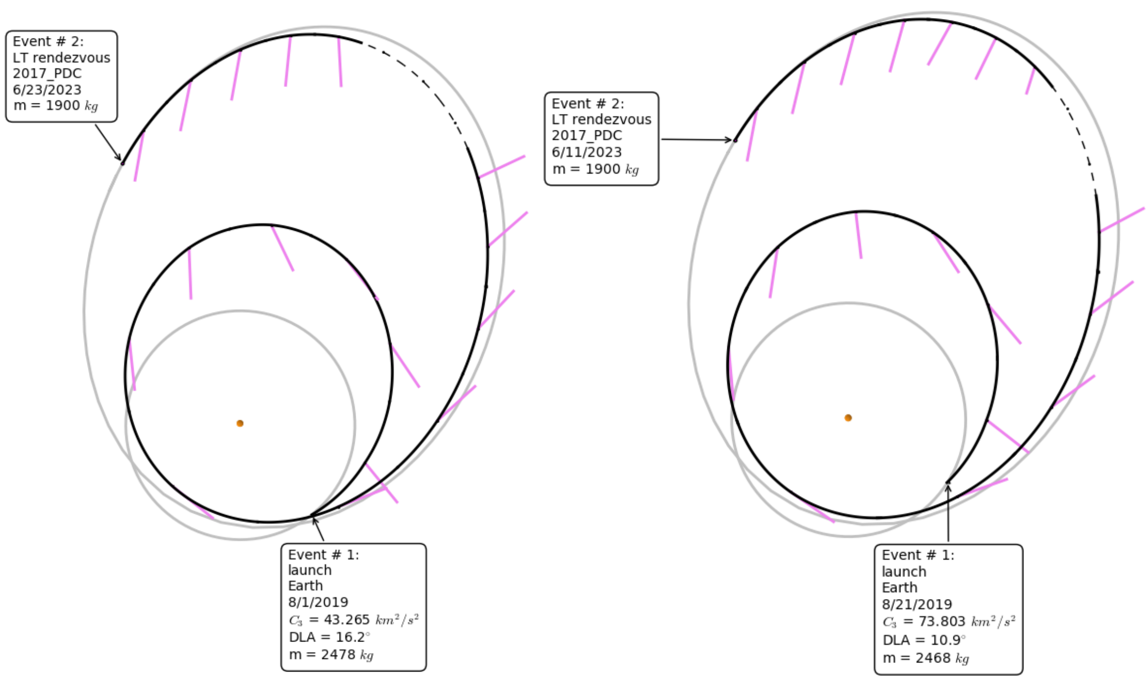
With a focus on current planetary defense efforts, this study developed a novel model for the asteroid's post-deflection trajectory. The purpose of this work was to generate a reasonably accurate solution for the body's conditions at Earth's close approach while being fast enough to be evaluated thousands or even millions of times, as would be needed in a mission's early conceptual stages. The model was demonstrated for mission concepts using either a Kinetic Impactor or Nuclear Explosive Device. With this new transcription of the post asteroid deflection, trajectory design as a whole can be quickly evaluated in an optimization tool. Finally, a trade study was performed for the fictitious asteroid impact scenario 2017 PDC, with optimized medium fidelity deflection dates and magnitudes confirmed by a high fidelity propagation of the asteroids post-deflection trajectory.



(a) Atlas V

(b) Delta IV Heavy

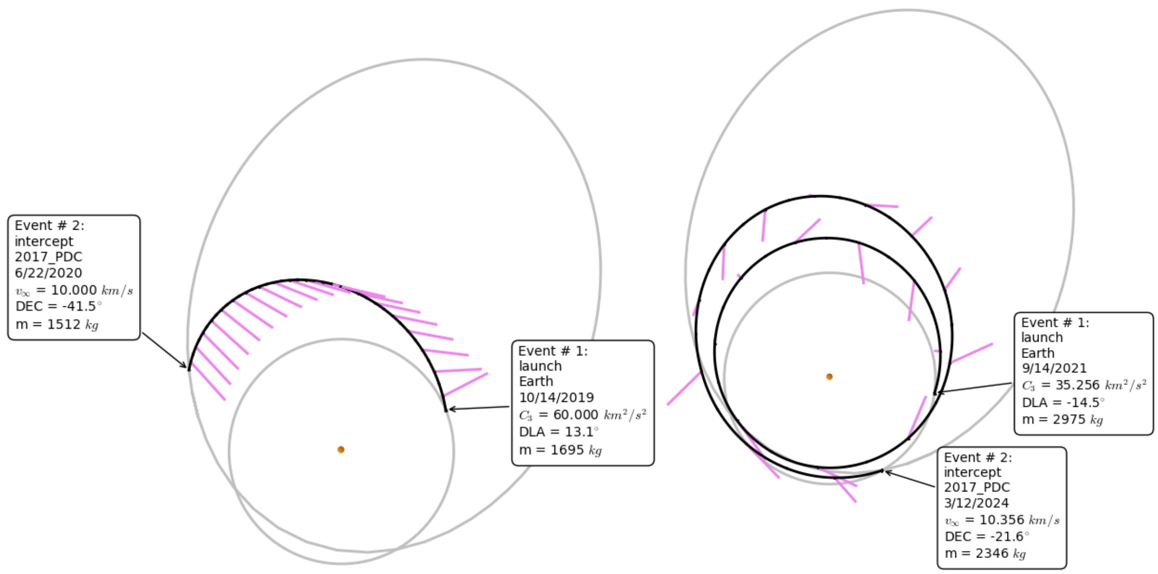
Fig. 4: Low-thrust single spacecraft with two flybys



(a) Atlas V

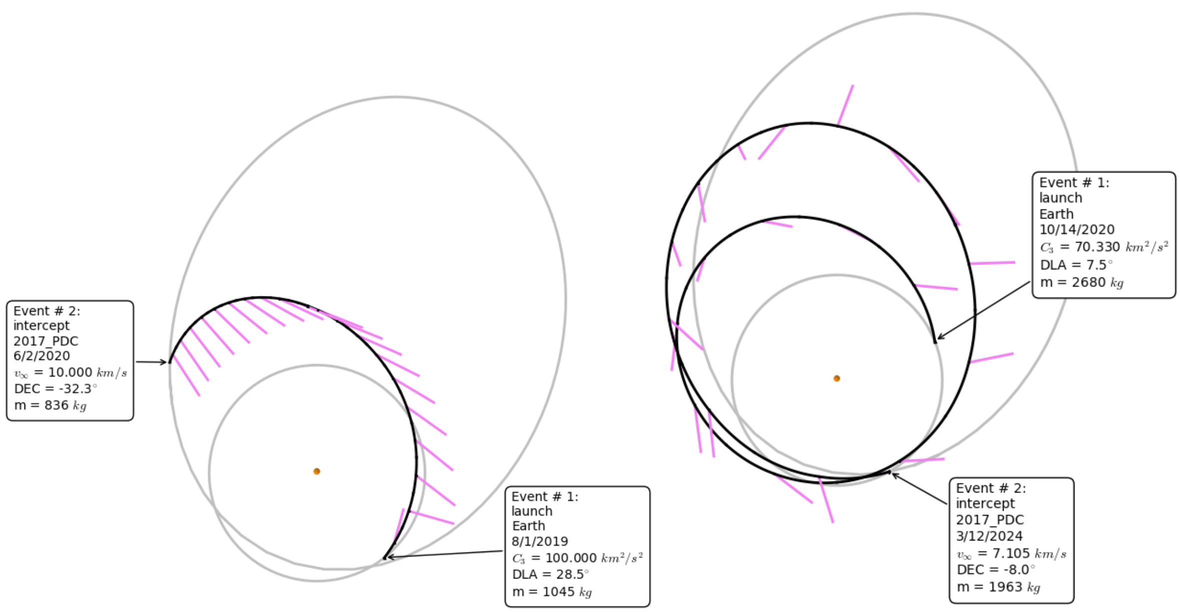
(b) Delta IV Heavy

Fig. 5: Low-thrust single spacecraft with rendezvous



(a) Survey (b) Deflection

Fig. 6: Low-thrust double spacecraft mission with Atlas V



(a) Survey (b) Deflection

Fig. 7: Low-thrust double spacecraft mission with Delta IV

References

[1] Popova, O. P., Jenniskens, P., Emel'yanenko, V., Kartashova, A., Biryukov, E., Khaibrakhmanov, S., ... & Badyukov, D. D. (2013). Chelyabinsk airburst, damage assessment, meteorite recovery, and characterization. Science, 342(6162), 1069-1073.

[2] Wie, B., & Barbee, B. W. (2015). An Innovative Solution to NASA's NEO Impact Threat

Mitigation Grand Challenge and Flight Validation Mission Architecture Development. Final Technical Report of a NIAC Phase 2 Study, December 9, 2014.

[3] Vasile, M., & Colombo, C. (2008). Optimal impact strategies for asteroid deflection. Journal of guidance, control, and dynamics, 31(4), 858.

[4] Vardaxis, G., & Wie, B. (2012). Asteroid mission design software tool for planetary defense applications. In AIAA/AAS Astrodynamics Specialist Conference, AIAA (Vol. 4872, p. 2012).

[5] Carusi, A., Valsecchi, G. B., D'Abramo, G., & Boattini, A. (2002). Deflecting NEOs in route of collision with the Earth. Icarus, 159(2), 417-422.

[6] Izzo, D. (2007). Optimization of interplanetary trajectories for impulsive and continuous asteroid deflection. Journal of Guidance Control and Dynamics, 30(2), 401-408.

[7] Englander, J. A., Conway, B. A., & Wall, B. J. (2009, May). Optimal strategies found using genetic algorithms for deflecting hazardous near-earth objects. In Evolutionary Computation, 2009. CEC'09. IEEE Congress on (pp. 2309-2315). IEEE.

[8] Casalino, L., & Simeoni, F. (2012). Indirect optimization of asteroid deflection missions with electric propulsion. Journal of Guidance, Control and Dynamics, 35(2), 423-433.

[9] Park, S. Y., & Ross, I. M. (1999). Two-body optimization for deflecting Earth-crossing asteroids. Journal of Guidance, Control, and Dynamics, 22(3), 415-420.

[10] Conway, B. A. (2001). Near-optimal deflection of earth-approaching asteroids. Journal of Guidance, Control, and Dynamics, 24(5), 1035-1037.

[11] Barbee, B. W., Sarli, B. V., Lyzhoft, J., Chodas, P. W., & Englander, J. A. (2017). Spacecraft Mission Design for the Mitigation of the 2017 PDC Hypothetical Asteroid Threat. 5th International Academy of Astronautics (IAA) Planetary Defense Conference 2017, 15-19 May 2017, Tokyo, Japan.

[12] Knittel, J., Hughes, K., Englander, J., & Sarli, B. (2017). Automated Sensitivity Analysis of Interplanetary Trajectories. 26th International Symposium on Space Flight Dynamics, 3-9 Jun. 2017, Maysuyama, Japan.

[13] Englander, J. A., Vavrina, M. A., & Hinckley, D. (2016). Global Optimization of Low-Thrust Interplanetary Trajectories Subject to Operational Constraints. AAS/AIAA Space Flight Mechanics Meeting, 14-18 Feb. 2016, Napa Valley, CA, United States.

[14] Board, S. S., & National Research Council. (2010). Defending planet earth: Near-Earth-Object surveys and hazard mitigation strategies. National Academies Press.

[15] Syal, M. B., Owen, J. M., & Miller, P. L. (2016). Deflection by kinetic impact: Sensitivity to asteroid properties. Icarus, 269, 50-61.

[16] Weaver, R. P., Barbee, B. W., Wie, B., & Zimmerman, B. (2015). Los Alamos RAGE Simulations of the HAIV Mission Concept. 4th IAA Planetary Defense Conference, 13-17 Apr. 2015, Frascati, Roma, Italy.

[17] Englander, J. A., Vavrina, M. A., & Ghosh, A. R. (2015). Multi-Objective Hybrid Optimal Control for Multiple-Flyby Low-Thrust Mission Design. 25th AAS/AIAA Space Flight Mechanics Meeting, 11-15 Jan. 2015, Williamsburg, VA, United States.

[18] Battin, R. H. (1999). An introduction to the mathematics and methods of astrodynamics. AIAA.

[19] Wilkinson, J. H. (1965). The algebraic eigenvalue problem (Vol. 87). Oxford: Clarendon Press. pp 440.

[20] International Academy of Astronautics (2017). 5th IAA Planetary Defense Conference. *http://pdc.iaaweb.org/*, May 15-19, 2017, Tokyo, Japan.

[21] NASA Launch Services Program. Launch Vehicle Performance Website. *https://elvperf.ksc.nasa.gov/Pages/Query.aspx*.